

FINAL REPORT EMERGENCY EARTH ORBITAL ESCAPE DEVICE STUDY

**VOLUME 4: ● APOLLO APPLICATIONS PROGRAM
● EMERGENCY ESCAPE SYSTEM
● PRELIMINARY PROGRAM DEFINITION PLAN**

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EMERGENCY EARTH ORBITAL ESCAPE DEVICE

Volume 4 Apollo Applications Program Emergency Escape System Preliminary Program Definition Plan

This document is the final report for the "Emergency Earth Orbital Escape Device Study" that was completed by Lockheed Missiles & Space Company for the National Aeronautics and Space Administration, under Contract NAS9-7907.

The final report has been prepared in volumes as follows:

- Volume 1 Condensed Summary
- Volume 2 General Technical Summary
- Volume 2A Systems Requirements and Concepts
- Volume 2B Spacecraft System Design
- Volume 2C Reentry Controls
- Volume 2D Environmental Control, Communications, and Electrical Systems
- Volume 2E Additional Study of Tasks
- Volume 3 Preliminary Program Definition Plan
- Volume 4 Apollo Applications Program Emergency Escape System
Preliminary Program Definition Plan

This study was completed under the direction of Mr. Ray Bradley, NASA/MSC Study Manager.

The study was managed and supervised by the following Lockheed Missiles & Space Company personnel:

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CONTENTS

	Page
1.0 INTRODUCTION	2
1.1 Objectives	2
1.2 Overview	3
2.0 SYSTEM CONCEPT	4
2.1 Launch Vehicle System	8
2.2 Rendezvous and Docking System	10
2.3 Escape System	12
2.4 Escape System Requirements and Configuration	14
2.5 Escape System Structure	16
2.6 Retro Propulsion and Pyrotechnics System	18
2.7 Reaction Control System	20
2.8 Environmental Control System	22
2.9 Electrical System	24
2.10 Communications System	26
2.11 Orbiting Beacon	28
3.0 PRELIMINARY PROGRAM DEFINITION PLAN	30
3.1 Program Management Plan	33
3.2 Reliability Program Plan	34
3.3 Engineering and Design Plan	35
3.4 Procurement Plan	37
3.5 Manufacturing Plan	39
3.6 Test Plan	43
3.7 Facilities Plan	45
3.8 Launch Base Operations	47
3.9 Preliminary Price Estimates	49

1.0 INTRODUCTION

1.0 INTRODUCTION

The Emergency Earth Orbit Escape Device Study developed the concepts and definition for a simple emergency earth orbit escape system which meets or exceeds all mission requirements by means of established technology and extensive use of available flight hardware. The study results are directly applicable to the Apollo Applications Program and the escape system can be provided by one of two means. The escape system can be launched with the space station or launched separately for rendezvous and docking with the space station.

An analysis of the tradeoffs and quantitative advantages of each method of providing the escape system is beyond the intent and scope of this document. The rendezvous concept is presented to provide a basis for evaluating and weighing the relative advantages of providing an escape system by either means.

Unmanned rendezvous and docking can be performed by means of the Titan IIIB Agena or Atlas/Agena vehicles with varying maneuver and payload margins. A Thorad/Agena boost vehicle is marginal and requires an ETR pad modification. Existing Agena program hardware provide the necessary on-orbit command and maneuver capability. The three-man escape system weight is 2500 pounds, and the alternate nine-man unit equipped for a three-man crew weighs 5200 pounds without the crew. Titan IIIB provides a 2200-pound payload margin plus maneuver capability for the nine-man unit and 4700 pounds for the three-man unit. Either the three-man or the nine-man unit, as defined, can be developed to support an August 1971 mission by means of the preliminary program definition plan presented. The recurring cost for the three-man escape is approximately \$3.3 million and the development cost is approximately \$26 million.

1.1 OBJECTIVE

The program objective is to provide an emergency escape system capability in support of the earth orbit Apollo Applications Program missions in 1971 and 1972.

1.2 OVERVIEW

In order to present a meaningful basis for evaluation of the program definition plan, the system concepts and configurations which are to be implemented by the plan are first presented and defined in as much detail as is pertinent to this document. The launch vehicle and rendezvous systems are presented only as they affect the overall system concept and the escape mission directly. The launch vehicle and rendezvous systems are defined in more detail by Lockheed report LMSC/A852985, Confidential, which is available as needed. Only one of many possible mission concepts and scenarios is presented. The many obvious options and probable mission applications will not significantly alter the basic overall system configuration or program definition plan.

The preliminary program definition plan presents the detailed engineering development, reliability, procurement, manufacturing, test and launch operations plans and schedules for implementing the escape system configurations and concepts presented. The plan provides an optional flight test three months before the scheduled August 1971 AAP space station launch. The optional flight test or demonstration unit is not specifically recommended or priced, but is included in the planning and scheduling as an optional item which may be deemed necessary, dependent upon the final mission concept and requirements.

Section 6.9 of the program plan presents the total program cost estimates for the program as defined and detailed. The system configuration and program plan have been designed to provide high reliability and minimum cost by means of studied and astute simplicity.

2.0 SYSTEM CONCEPT

The system provides a simple, reliable escape system for emergency egress and return to earth from the AAP space station.

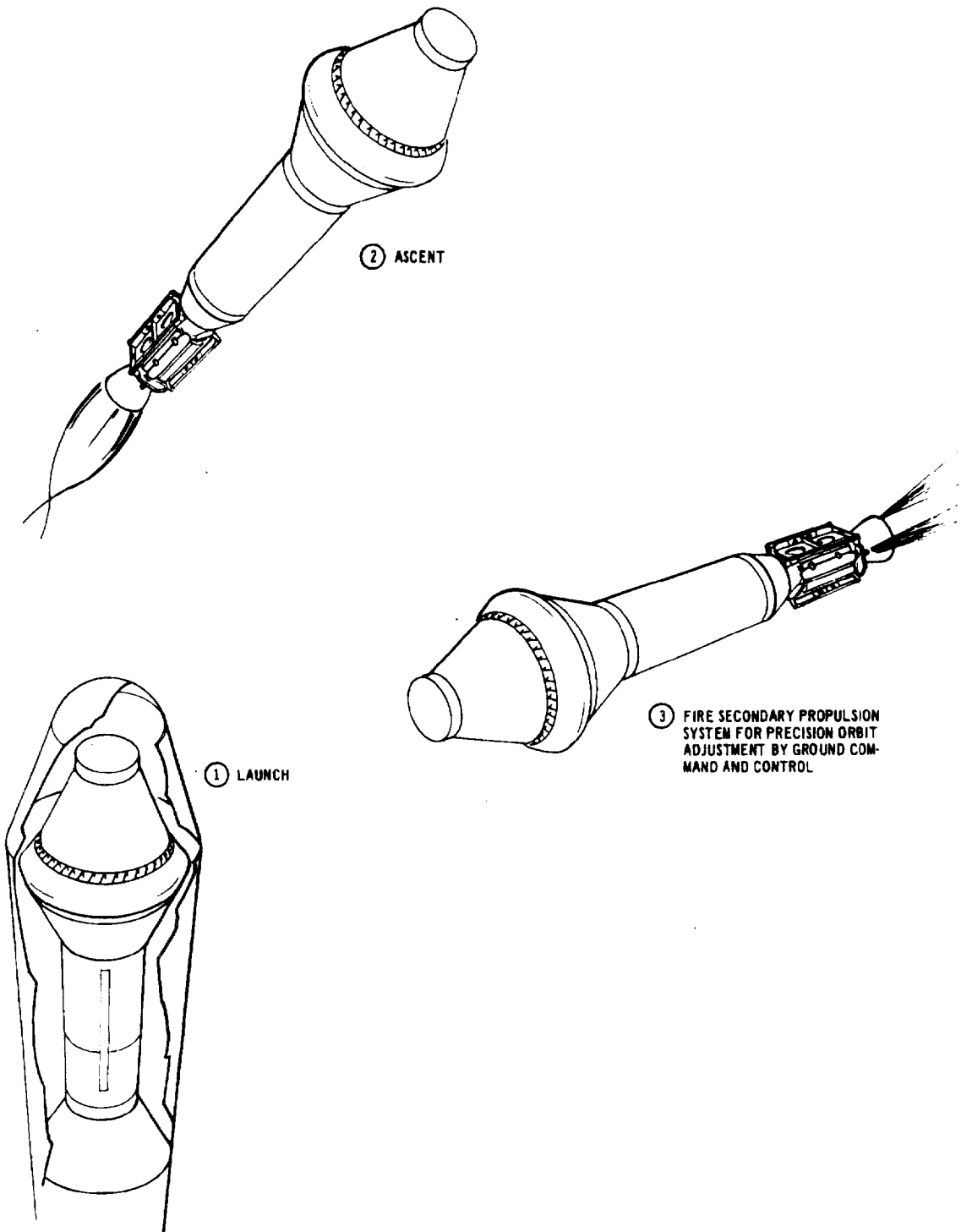
The system contains a minimum of components and complexity and makes maximum possible use of available technology and hardware. Relatively simple manually controlled systems are used to provide high reliability and low cost. The overall system is divided into the following parts:

- Titan IIIB launch vehicle system
- Agena, rendezvous and docking system
- Emergency escape system

In order to keep the emergency escape system simple, use is made of the Titan IIIB and Agena for launch, rendezvous and docking of the escape system as a passive payload. Once the escape system is docked, the Agena is separated and used for other purposes or moved out of the space station orbit trajectory. The escape system hatch will be opened and the system checked out by the astronauts and statused, at which time it will be ready for emergency escape.

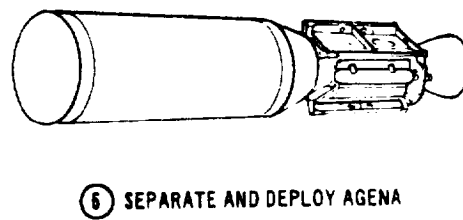
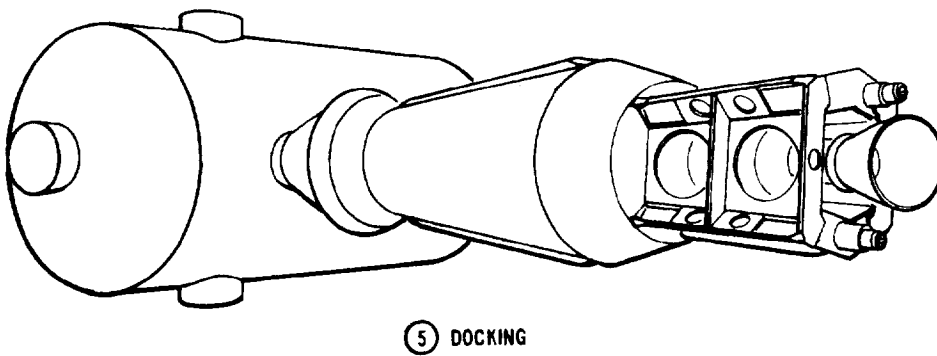
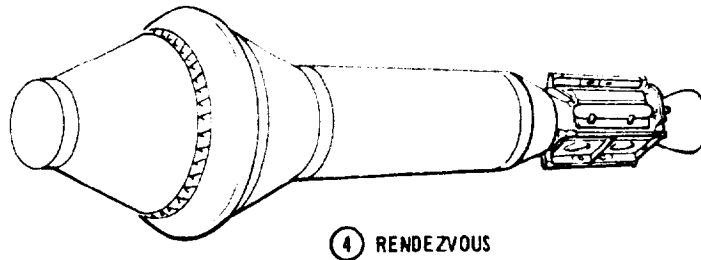
When used for escape, the astronauts enter the system, secure the hatch, activate the environmental control and reaction (attitude) control, system and separate from the space station by ignition of a pyro or "V" band separation joint in the docking adapter. Separation action and pressure with the docking adapter will pulse the vehicle away from the station and may impart a slow tumble of a few degrees per second. The artificial horizon and yaw indicator is a simple self-contained instrument containing a rate mode for use in the event of a tumble condition. The pilot aligns the vehicle to the earth's horizon by means of a sight and reticle on the hatch window, cages and uncages the artificial horizon indicator and sets the earth geocentric rate potentiometer. Re-entry time is verified by radio contact with the ground. The astronauts may loiter on orbit for up to nine to twelve hours while preparations are made for their recovery from the water and their orbit precession carries them into a select landing area. The pilot alone controls the vehicle attitude. He puts the escape vehicle into the proper

ASCENT SEQUENCE



attitude for retro propulsion by reference to the earth horizon or artificial horizon indicator and fires the retro rockets. After retro, the vehicle is oriented for atmospheric reentry such that the system will have a minimum angle of attack. But if the vehicle enters the atmosphere backwards, the aerodynamic stability will right the system. At 400,000 feet the pilot separates the retro module and reaction control system. At approximately 80,000 and Mach 2, the drogue chute is deployed and is later separated and a reefed main chute deployed. After water impact the main chute is separated and the system floats upright on the heat shield due to hydrodynamic single-point stability. The vehicle will right itself if inverted by wave action. If necessary, the astronauts can open the hatch and exit the escape vehicle. The battery supply and fan allow the astronauts to remain in the escape vehicle for at least twenty four hours with the hatch secured.

RENDEZVOUS AND DOCK SEQUENCE



2.1 LAUNCH VEHICLE SYSTEM

The launch vehicle consists of the Titan IIIB Agena and emergency escape system.

The escape system will be launched from the Kennedy Space Center, rendezvous, and dock with the space station. The orbit adjust capability requirements are dependent upon space station orbit parameters, injection accuracy, timing of the escape system launch, and the coast or on-orbit maneuver time capabilities of the rendezvous vehicle. The longer the on-orbit time allowed for orbit phase adjustment between the space station and rendezvous, the smaller the required adjust velocity for rendezvous. An orbit adjust velocity capability of 1000 ft/sec for plane, altitude and phase adjustment has been used for evaluation of the capabilities of various boost vehicles.

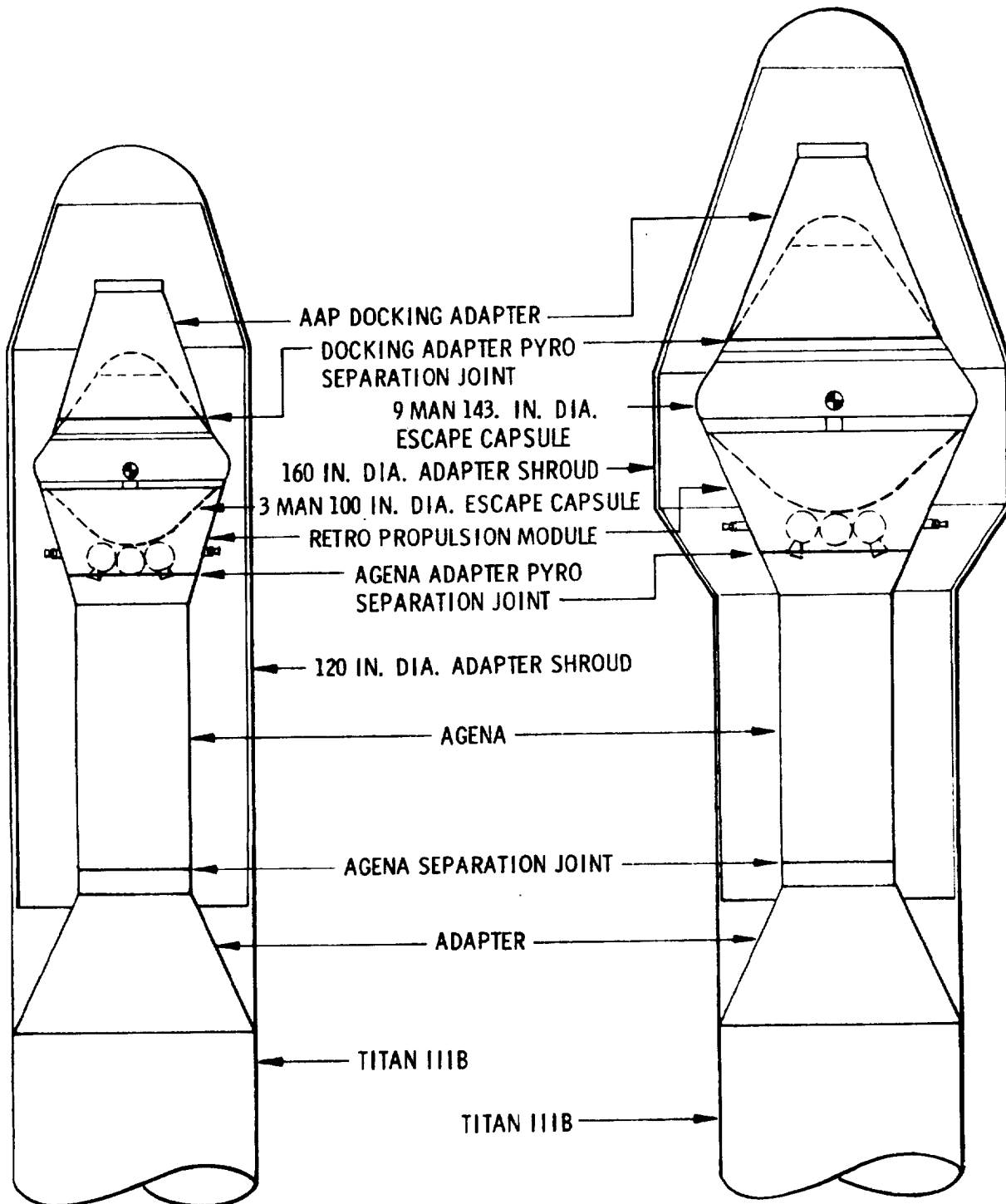
Payload capabilities of three candidate boost vehicle systems for injecting the escape system into 210 mile altitude orbit at 35 degrees inclination are as follows:

- | | |
|------------------------|---------------|
| • Atlas SLV-3C/Centaur | 10,600 pounds |
| • Atlas SLV-3A/Agena | 7,500 pounds |
| • Titan IIIB/Agena | 8,300 pounds |
| • Thorad/Agena | 3,400 pounds |

The Titan IIIB/Agena is recommended for use as the boost vehicle due to its payload and launch response time capabilities and availability of the Integrate, Test and Launch complex. An existing program is flying Titan-launched Agenas with secondary propulsion, command and control capabilities analogous to the Gemini Agena Target Vehicle capabilities*. The Titan IIIB Agena scheduled to launch the Comsat Corporation Intelsat IV satellite from Kennedy Space Center in 1970 will provide the required ten-foot shroud and launch capabilities. Launch vehicle configurations for the three-man and nine-man systems are shown on the adjoining page. The nine-man system requires the development of a 160-inch diameter hammer head shroud as shown. The excess payload capability of the Titan IIIB/Agena is approximately 4,700 pounds for the three-man system and 2,200 pounds for the nine-man system.

*Details of the vehicle, hardware and rendezvous are presented in Lockheed report LMSC/A852985, March 1967, Confidential.

TITAN IIIB/AGENA LAUNCH VEHICLE



2.2 RENDEZVOUS AND DOCKING SYSTEM

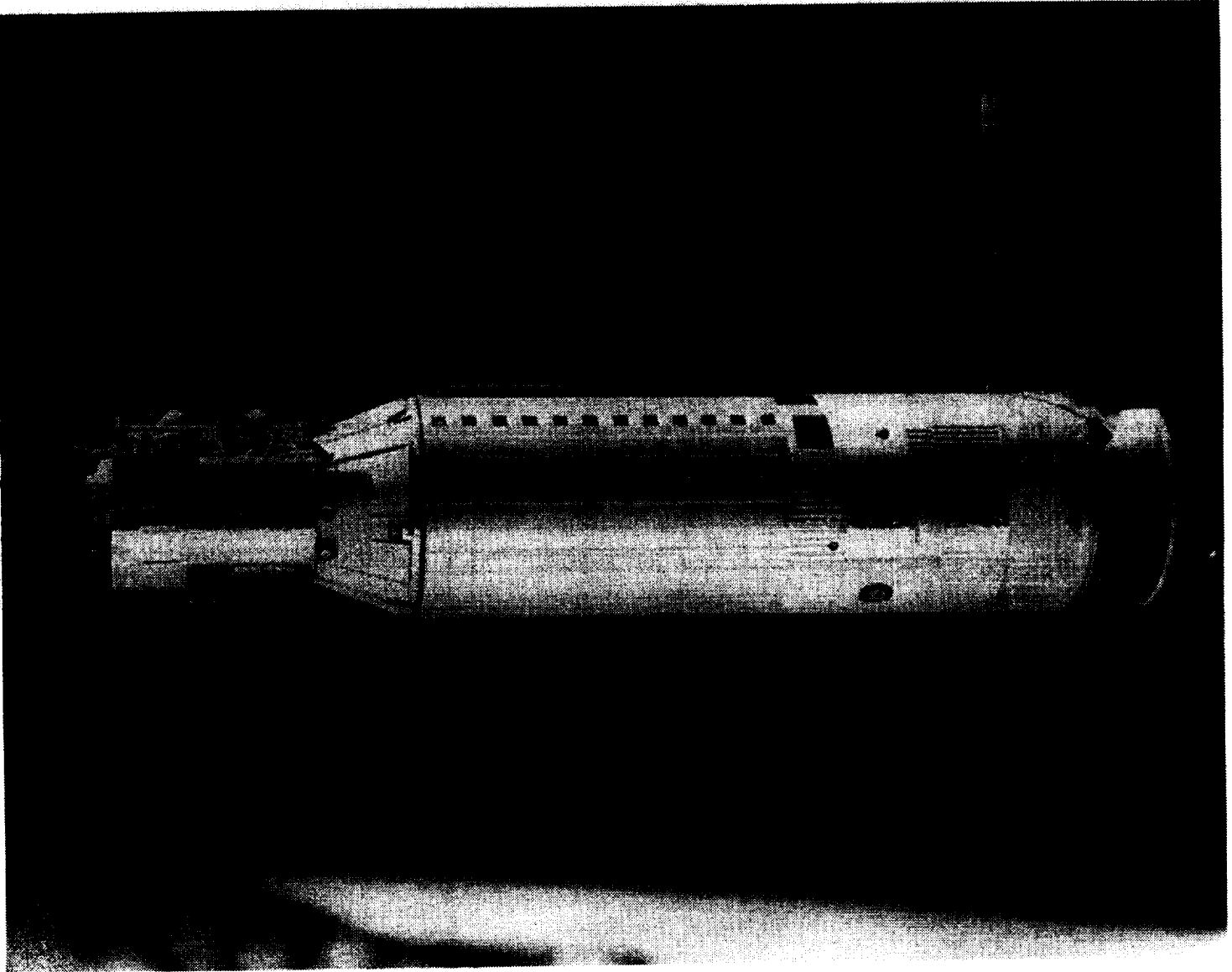
Rendezvous will be made from a direct ascent. Docking will be performed by remote control from the space station.

Escape system launch will be at a precise time and on a precise injection trajectory which will effect rendezvous with a minimum of ground controlled orbit adjustment. The launch vehicle system utilizes storable propellants and has demonstrated reliable on-time launch response. Rendezvous maneuvers will be performed by means of ground tracking and control. Once the rendezvous vehicle is within control range, final rendezvous and docking will be controlled by the astronauts. The rendezvous vehicle will be able to maintain a controlled attitude to within a fraction of a degree by means of the inertial reference package and autopilot. Position translation movement will be controlled by means of a set of low-level (ten-pound) thrusters acting essentially through the vehicle center of gravity. Dispersion of the center of gravity from the thrust vector axes is compensated for by the attitude control thrusters.

After docking of the escape system, the Agena is separated by means of a pyro joint and commanded out of the workshop orbit.

An existing Agena program has developed a secondary propulsion system and a command control unit for precise orbit adjust. Additional equipment which is compatible with the NASA tracking network is available from the Gemini Agena Target Vehicle shown on the adjoining page.

GEMINI AGENA TARGET VEHICLE



2.3 ESCAPE SYSTEM

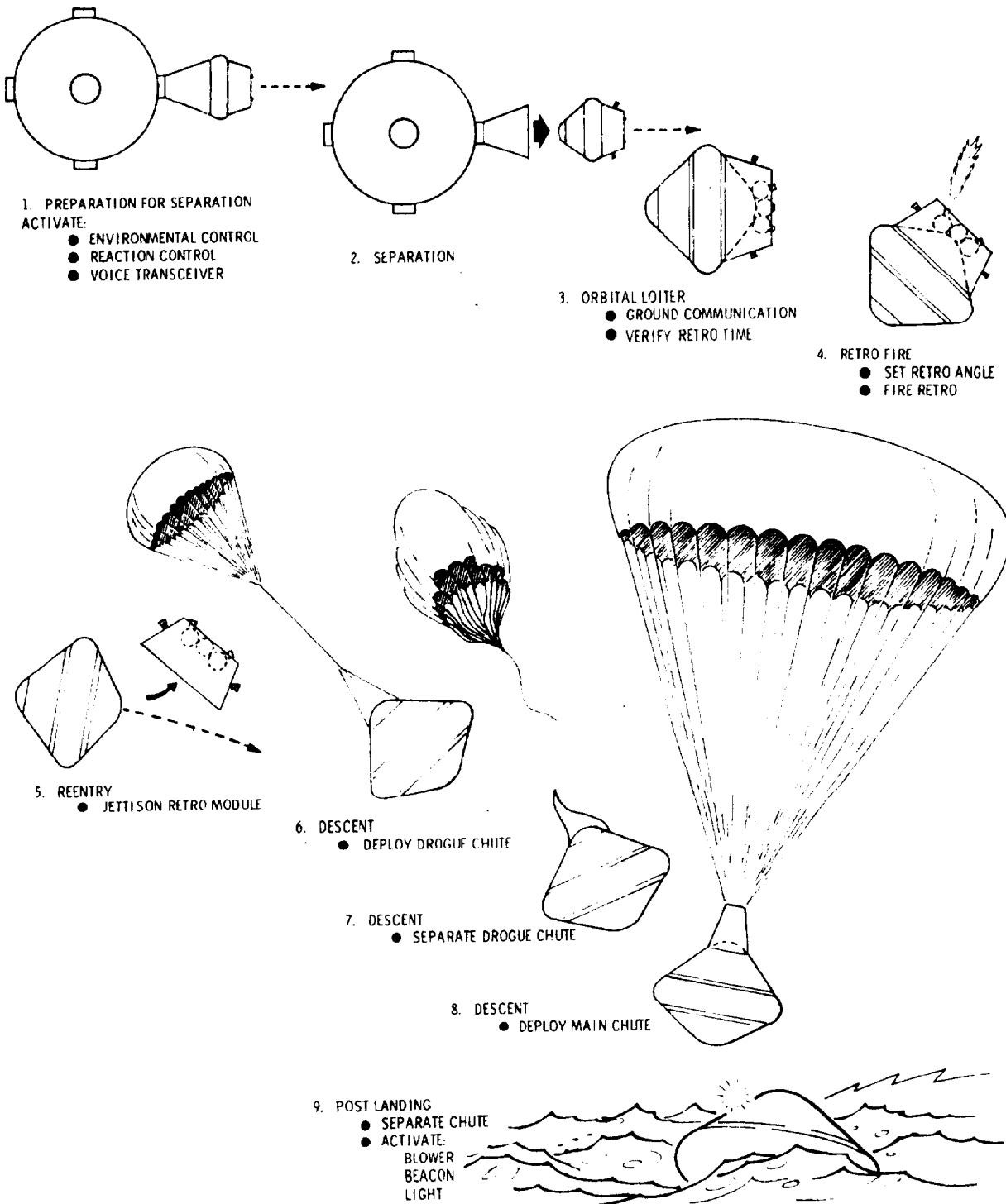
The system provides an escape capability that is ready for immediate use at any time.

A 9-hour orbit loiter capability provides time for ground communication prior to retro and permits orbit precession to allow selective landing in an area within normal flight range of air-sea rescue planes.

The mission sequence of events is as follows:

1. Hatch closed; activate environmental and reaction control systems; separate from the space station by ignition of docking adapter pyro joint.
2. Determine orbit attitude by sight and reticle on the hatch window; manually control attitude; null artificial horizon with the visual reference attitude.
3. Verify time for retro fire through ground link voice communication; orient vehicle to the retro attitude; initiate retro fire by switch located on the attitude hand controller.
4. Maintain retro attitude throughout retro fire; jettison retro module at approximately 400,000 feet.
5. Deployment and separation of the drogue and main parachute systems by the recovery timer with manual back-up switches.
6. Impact and main parachute separation; beacon light activated; location aids deployed; post-landing exhaust fan turned on.

ESCAPE SEQUENCE



2.4 ESCAPE SYSTEM REQUIREMENTS AND CONFIGURATION

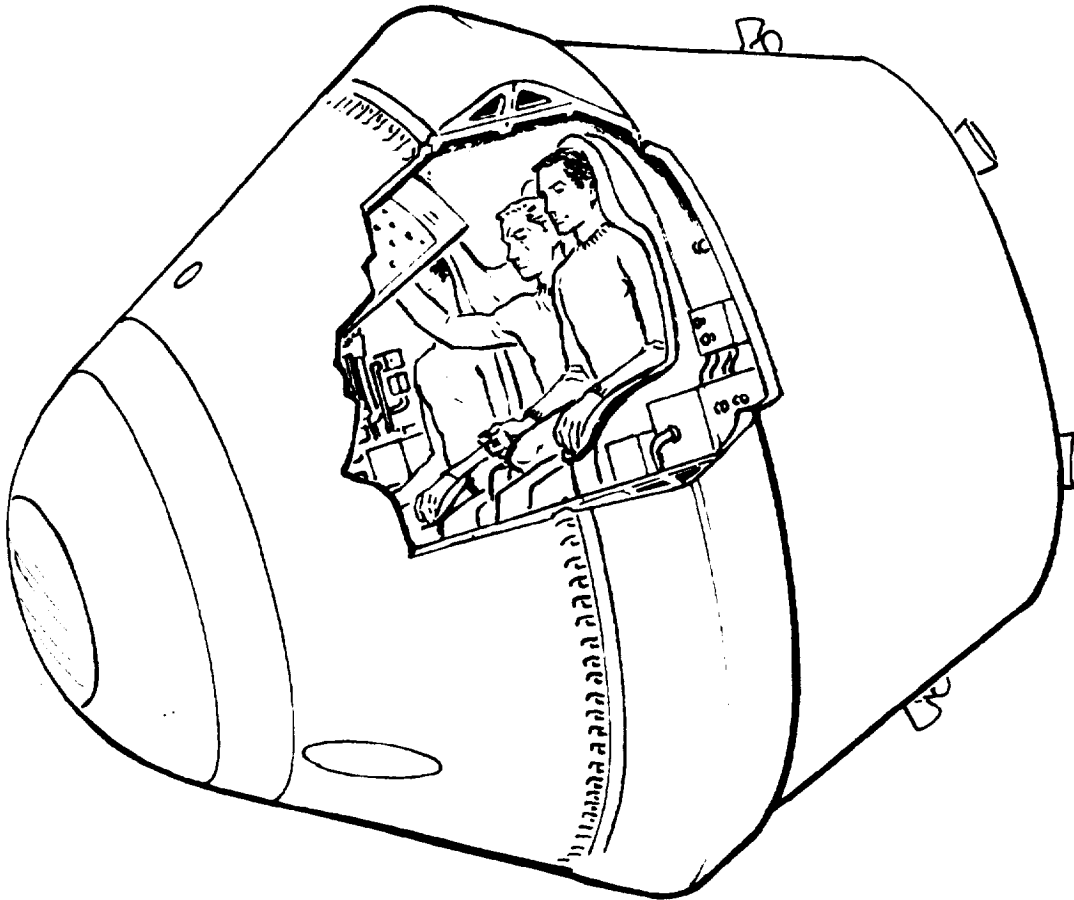
The system configuration satisfies the Apollo Applications escape mission requirements.

The phase A and B study requirements which dictated the system configuration, encompass the Apollo Applications escape system requirements as follows:

<u>Study Requirements</u>	<u>Apollo Applications Requirements</u>
100 - 300 nm orbit altitude	210 nm orbit altitude
28- to 90-degree orbit inclination	35 degrees
12-hour on-orbit loiter time	9 hour maximum
Water landing only	Same
24-hour minimum post-landing electrical capability	Same
Ballistic (zero life) aerodynamic re-entry	Same
Single-point aerodynamic stability	Same
Single-point flotation stability	Same
1-, 2-, or 3-man optional escape crew	Same
Suited or unsuited ingress	Same
Suit removal capability	Same
Single-gas environmental control system	Same
Near-zero physiological angle	Same
Five-year orbit storage life	9 month on-orbit life

The resulting three-man system configuration weight is 2400 pounds stored on-orbit, and the size is 100 inches diameter by 100 inches long. The nine-man system weight is approximately 5200 pounds, and the size is 143 inches diameter by 143 inches long. The ablative heat shield is a modified 80-degree conical shape of 1.00 to 0.25 inch thickness of silicon elastomer contained in fiberglass honeycomb cells. Attitude is determined by visual reference. A hand control operates the monopropellant hydrazine reaction control system. Four 780-pound thrust solid propellant propulsion motors are fired sequentially for the three-man system to provide a 600 ft/sec retro velocity for re-entry. Drogue and main chutes provide final descent velocity control. The basic capsule structure is of aluminum ring and skin construction.

ESCAPE SYSTEM CONFIGURATION



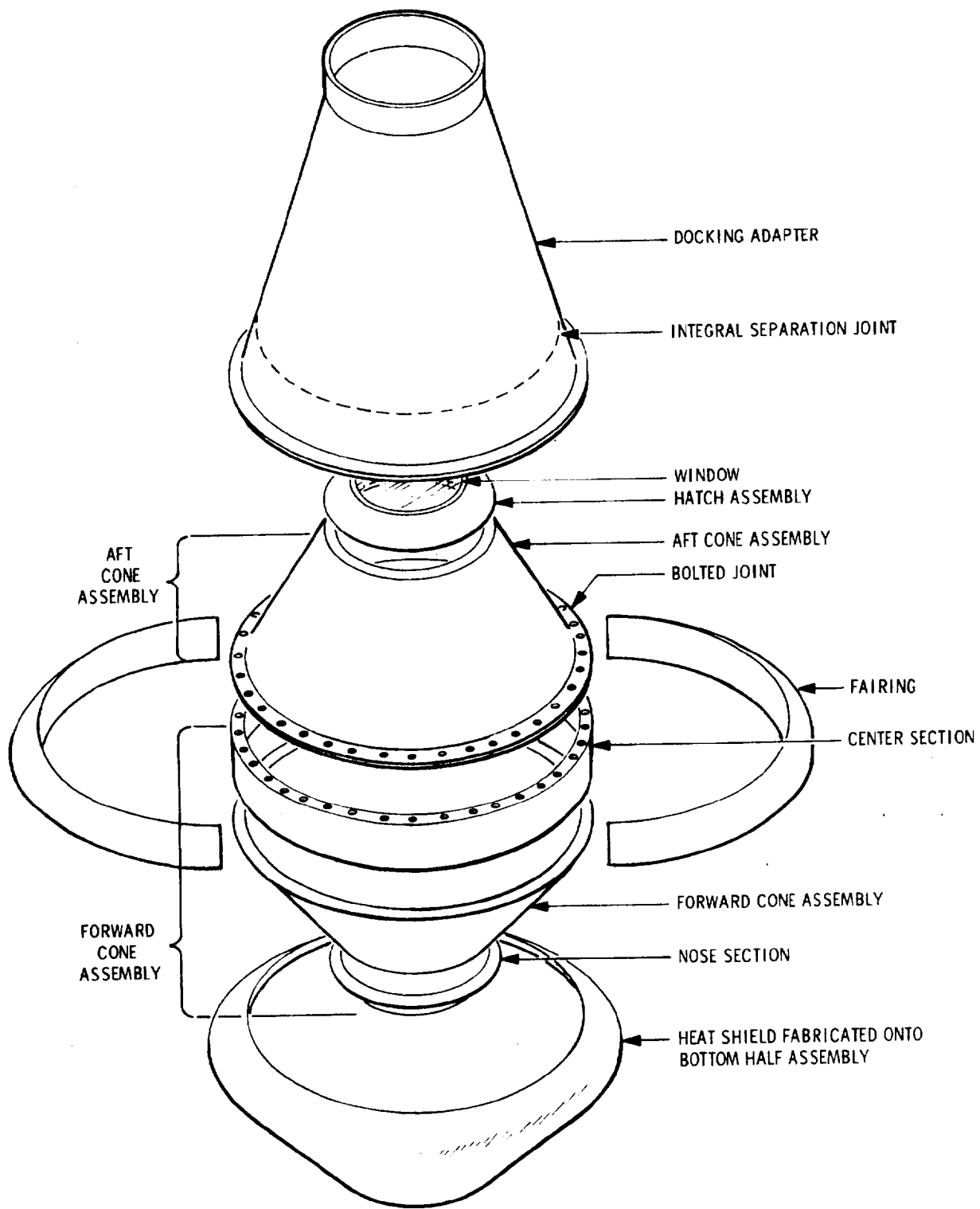
2.5 ESCAPE SYSTEM STRUCTURE

The structure is made up of conical and cylindrical sections with spheroidal ends.

Trade-off studies and analysis show that the aluminum structure provides minimum weight and minimum cost by present state-of-the-art manufacturing techniques. The structure is composed of welded skin and ring sections as illustrated on the adjoining page. A fairing around the short cylindrical section provides the proper aerodynamic and heat shield shape. The heat shield material is the same as used for the Gemini spacecraft. The propulsion module is attached to the vehicle by means of ablative stanchions protruding into pockets in the fairing and integrated into the heat shield. The propulsion module separation mechanism is similar to that used for NASA spacecraft, i.e., "V" shoes, strap, and ejection springs.

The bolt joint, as shown, separates the structure into two halves to facilitate manufacture and installation of equipment and subsystem modules. The docking adapter is joined to the escape system at the bolt joint to facilitate manufacture, inspection, test, and installation. The docking adapter and Agena adapter are separated by means of mild detonating fuse pyro joints. The drogue and main parachute assemblies are modularized for replacement while on orbit from within the pressurized work area.

STRUCTURAL ASSEMBLY



2.6 RETRO PROPULSION AND PYROTECHNICS SYSTEM

The retro propulsion and pyrotechnic systems are designed for simplicity and reliability.

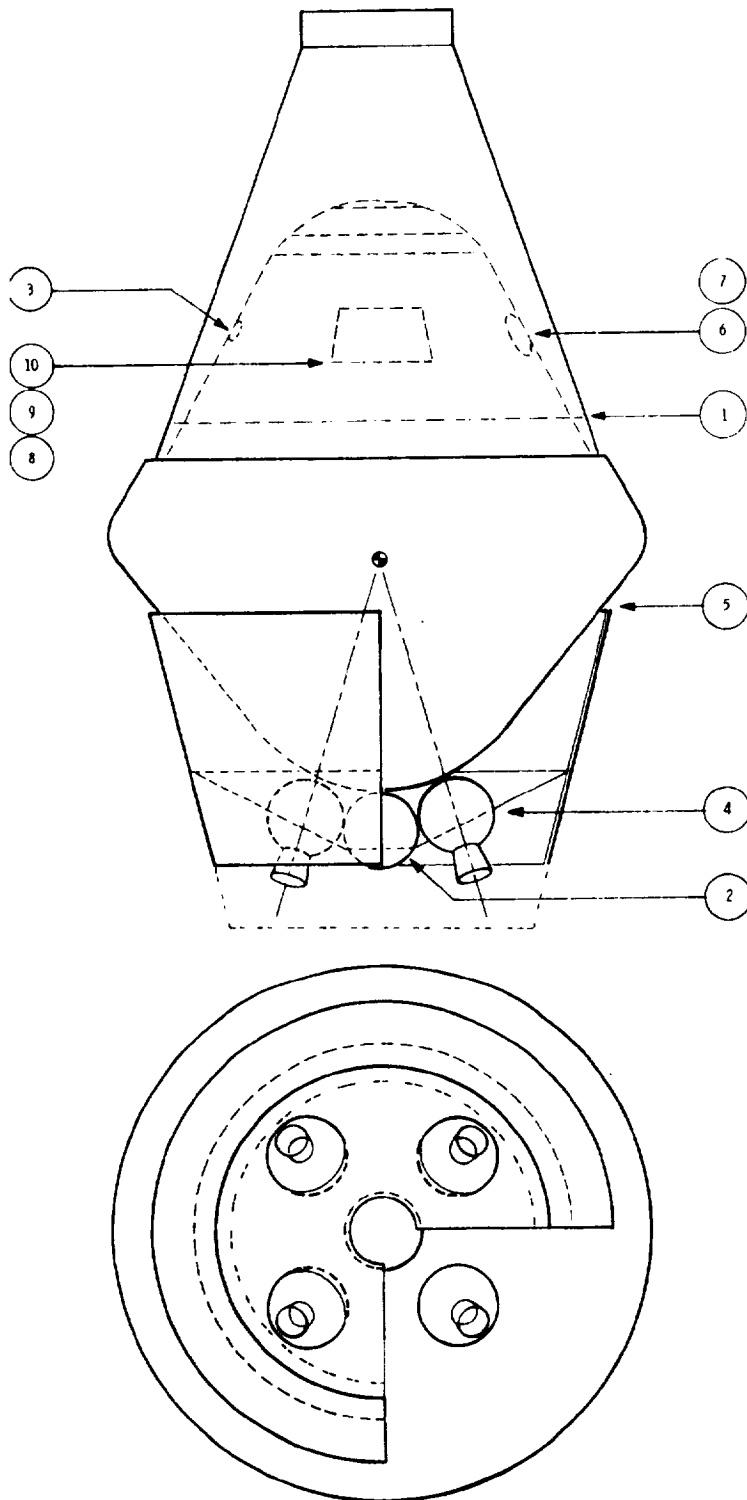
Retro propulsion for the three-man unit consists of four Thiokol 780-lb thrust TE 375 solid propellant motors which provide over 600 ft/second retardation. The motors are "off-the-shelf" items used on the Intelsat II and III satellites and have the following basic specifications:

- Length - 22.6 inches
- Diameter - 13.5 inches
- Weight - 77.6 lb
- Thrust - 780 lb
- Total impulse - 17,300 lb sec
- Total system weight - 312.0 lb

The nine man unit uses a larger size rocket motor. The minimum motor temperature is controlled during on-orbit storage and the orbital loiter period by an electric heater blanket and temperature control device for each motor. Retro propulsion ignition is manually controlled by means of a switch on the hand attitude controller. The motors are fired sequentially at 25-second intervals with about 5-second time delays between the thrust decay of one unit and the firing of the next unit.

The functions listed on the adjoining page are initiated by manually operated switches located on the control and display panel. Pyrotechnic redundancy is provided by dual igniters or squibs for each event. Each squib or igniter has two bridge wires. Each bridge wire is capable of firing the device.

PROPULSION AND PYROTECHNIC FUNCTIONS



1. STATION SEPARATION BY SQUIB-ACTUATED PRIMACORD JOINT
2. REACTION CONTROL SYSTEM ACTIVATION BY SQUIB-OPERATED ISOLATION VALVE
3. ORBITING BEACON SEPARATION BY SQUIB-OPERATED PINPULLER
4. RETRO MOTOR IGNITION BY SQUIB IGNITER
5. RETRO MODULE SEPARATION BY EXPLOSIVE BOLT & "V" BAND
6. DROGUE DEPLOYMENT BY MORTAR
7. DROGUE SEPARATION BY EXPLOSIVE NUT
8. MAIN CHUTE DOOR RELEASE BY SQUIB-OPERATED PINPULLER
9. MAIN CHUTE DE-REEF BY LINE CUTTER
10. MAIN CHUTE SEPARATION BY EXPLOSIVE NUT

2.7 REACTION CONTROL SYSTEM

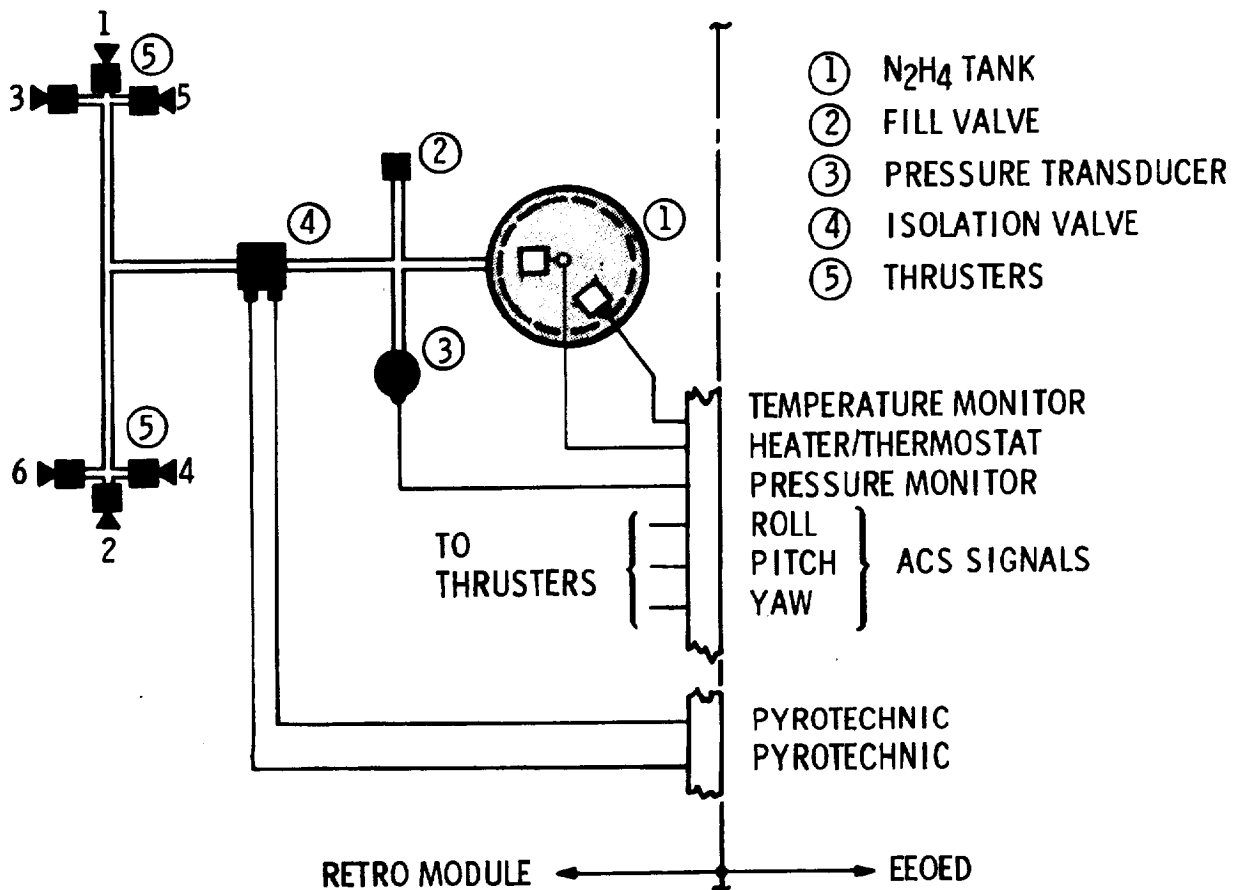
Manually controlled hydrazine monopropellant thrusters provide escape system attitude control.

Six thrusters provide pitch, yaw, and roll attitude control. One pair of thrusters provide pitch control, and the other four thrusters provide yaw and roll control. The on-off type thrust control values are rate and pulse width modulated to provide proportionate control. The modulation control electronics will be similar to building block units used on existing programs.

The titanium alloy propellant tank is available from an existing program and is sized to provide a total impulse of 5000 lb-sec plus ullage to assure a pressurization to decay ratio of 2. Positive propellant expulsion is achieved by the use of a surface tension device which maintains propellant at the outlet of the tank during low g conditions.

A single fill/drain valve is used for filling the system with the pressurant (nitrogen) and propellant (hydrazine). A squib actuated, normally closed, isolation valve is used to prevent leakage through the thrusters during the orbital storage period. The system uses all welded or brazed fittings to prevent leakage. Pressure and temperature transducers are used to indicate propellant quantity and to detect leakage. A propellant tank heater blanket provides thermal control during orbital storage. Two different spacecraft programs using comparable hydrazine reaction control systems are presently under development. "Off-the-shelf" flight-proven attitude control hardware is available. Due to the critical nature and characteristics of the reaction control system, it is strongly recommended that redundant systems be provided.

REACTION CONTROL SYSTEM DIAGRAM



2.8 ENVIRONMENTAL CONTROL SYSTEM

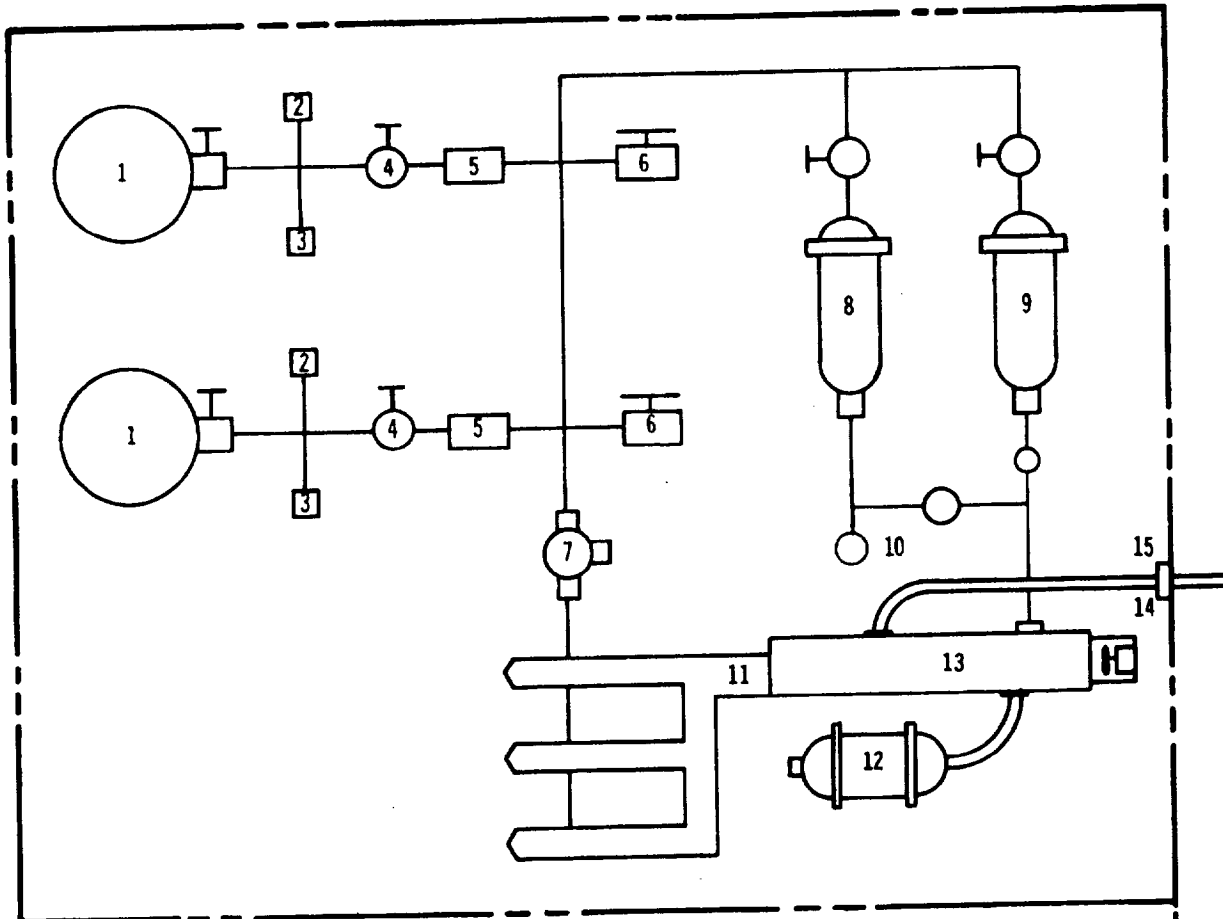
On-orbit environmental control is provided by a modularized closed loop system.

The system does not provide environmental control for suited crewmen and therefore pressure suits are removed after environmental system activation and cabin purging. Suit environmental control units would have increased the weight and complexity of the escape system.

Changeover from the two-gas, 7-psia station atmosphere to the single-gas, 5-psia escape system atmosphere will be accomplished after hatch closure by lowering the internal pressure to 6 psia and then purging the capsule with pure oxygen. The pressure will then be allowed to decay to 5 psia and maintained at that level until atmospheric air can be used after splash down.

The post landing environment is controlled by a fan which circulates cabin air from the outside. The environmental subsystem flow diagram shown on the adjoining page is similar to the system used for the Gemini spacecraft and is packaged into a single package module which will be tested and installed into the escape system with a minimum of plumbing and connection work. Two oxygen bottles are provided for redundancy against the possibility of a slow leak on orbit. The heat exchanger and boiler provide heat and moisture control during the orbit loiter period. The LiOH/charcoal filter removes contaminants and odors. Oxygen masks are included as an added emergency backup for short term use.

ENVIRONMENTAL CONTROL SYSTEM



WEIGHT _____ 100 LB
POWER, ORBIT _____ 20 W

1. O₂ SUPPLY
2. RELIEF VALVE
3. O₂ FILL VALVE
4. SHUTOFF VALVE
5. PRESSURE REDUCER
6. PRESSURE REGULATOR, 5 PSIA
7. O₂ RATE CONTROL VALVE
8. DRINKING WATER
9. BOILER SUPPLY WATER

10. WATER VALVE
11. O₂ DISTRIBUTION DUCTING
12. LiOH-CHARCOAL FILTER
13. GAS/H₂O HEAT EXCHANGER
14. CABIN FAN
15. STEAM VENT
16. POST-LANDING EXHAUST FAN
17. CABIN RELIEF VALVE, 6 PSIA

2.9 ELECTRICAL SYSTEM

All electrical power is provided by two long-life rechargeable batteries.

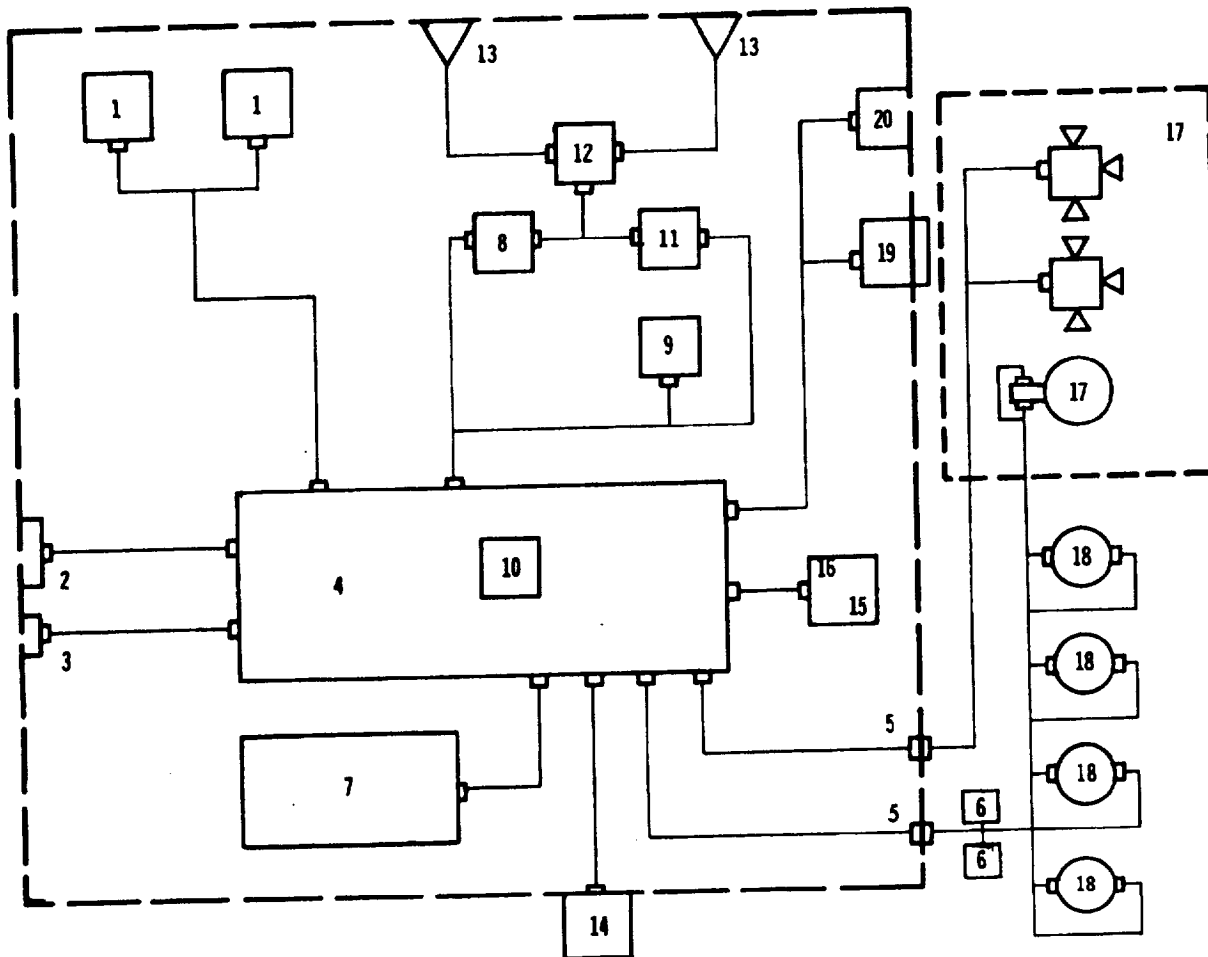
Twenty eight volt dc electrical power is provided by means of two 105 ampere hour silver-zinc secondary batteries sized to meet the mission peak load and long term power requirements. Battery power is cabled directly to the power distribution and control unit which is integrated with the control panel. Electrical systems are controlled by switches located on the control panel.

Each battery is kept on a trickle charge from station power through the umbilical to insure that the batteries will remain fully charged. Each battery is protected by means of isolation diodes and a circuit breaker which can be reset. The power system block diagram is shown on the adjoining page.

All cabling is sheathed cable or contained within a cabling sheath. All cable connectors are potted. All boxes are fully enclosed with pressure relief valves and all connections flow-coated and insulated with epoxy or silicone rubber. All switches are pressure tight. The switches are mounted within the pressure distribution and control box with the control knobs protruding through the control panel face. Rubber boots are sealed to the control panel face and protrude over the switch knobs to provide a positive sealing barrier not dependent upon sliding or rolling component fits.

Pseudo rate control electronics for modulation of the reaction control valves are packaged into a replaceable electronic module within the hand control box. Separate modules without relays or switches are provided for each set of redundant reaction control valves.

ELECTRICAL SYSTEM BLOCK DIAGRAM



- 1 BATTERY
- 2 UMBILICAL
- 3 SEPARATION IGNITER
- 4 POWER DISTRIB AND CONTROL PANEL
- 5 SEPARATION PLUG
- 6 PYRO SEPARATION
- 7 ENVIRONMENTAL CONTROL
- 8 VHF BEACON
- 9 VHF TRANSCEIVER
- 10 AUDIO UNIT

- 11 MULTIPLEXER
- 12 POWER DIVIDER
- 13 ANTENNA
- 14 ORBITING BEACON
- 15 HAND CONTROLLER
- 16 PSEUDO RATE CONTROL
- 17 REACTION CONTROL SYSTEM
- 18 RETRO MOTORS
- 19 RECOVERY SYSTEM
- 20 FLASHING LIGHT

2.10 COMMUNICATIONS SYSTEM

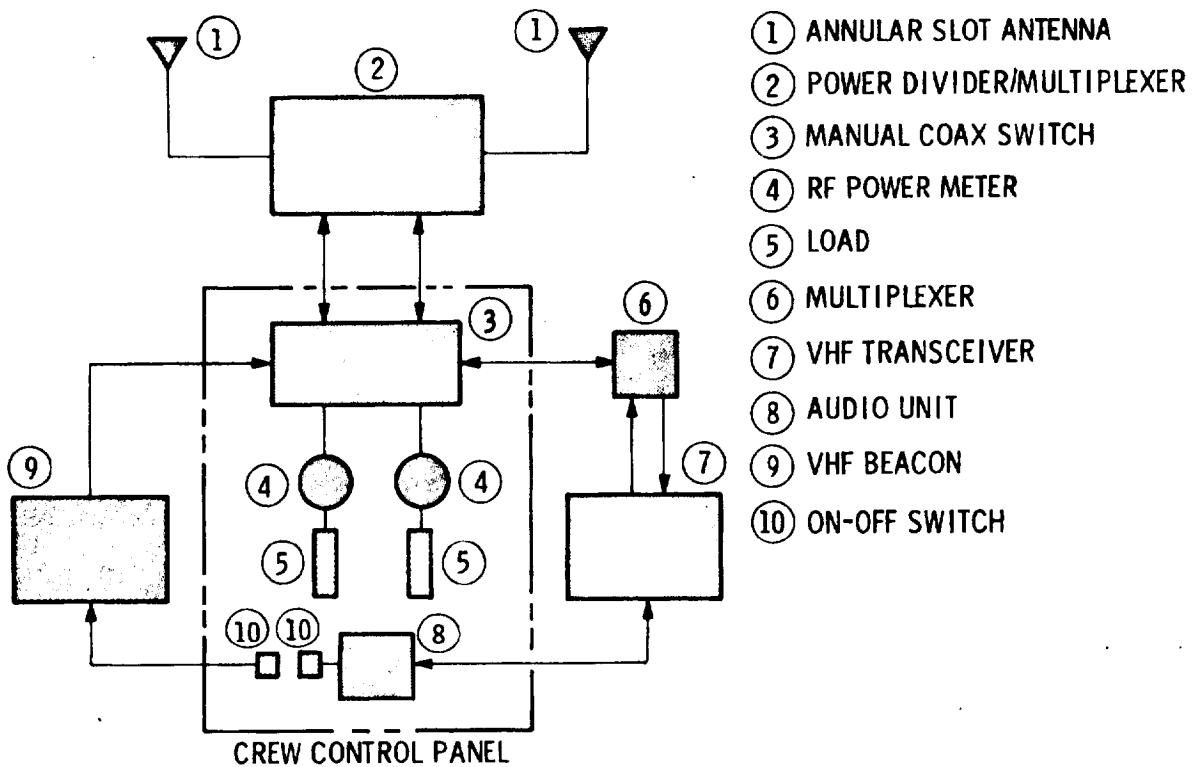
Communications consist of a VHF transceiver for voice communication and a VHF recovery beacon, operating at the international distress frequency 243 MHz.

The voice system will be used to transmit astronaut crew status and to receive ground control instructions for retro time.

Output from the transceiver is fed through a multiplexer to a manually operated coaxial switch located on the control panel. Output from the beacon is also fed to the switch providing on-board checkout capability for both the transceiver and the beacon through the RF power meters located on the control panel. In the operational mode, both beacon and transceiver outputs are switched through the power divider/multiplexer to flush-mounted annular slot antennas located on the aft cone structure.

The communications subsystem block diagram is shown on the adjoining page. All items shown in the block diagram except the antennas and audio unit have been developed and flight-proven on the Gemini spacecraft and other programs.

COMMUNICATIONS SYSTEM BLOCK DIAGRAM



2.11 ORBITING BEACON

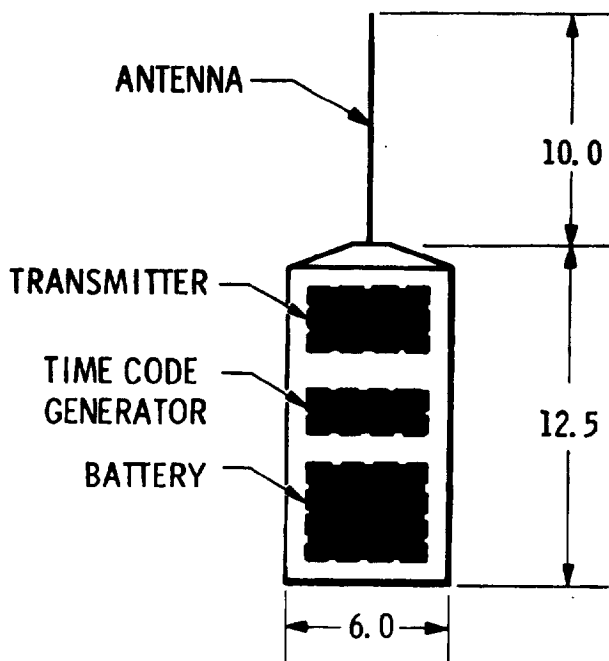
An ejectable beacon is left in orbit to signal the time of retro initiation and orbit as an aid to search and rescue.

The beacon, consisting of a 2-watt transmitter, binary time code generator, and power supply, will be capable of transmitting for 24 hours after ejection from the escape system. The antenna is an unfurlable whip made from beryllium copper or gold plated stainless steel.

The beacon configuration shown on the adjoining page is designed mainly from available integrated solid state modules.

ORBITING BEACON CONFIGURATION

- EJECTED AT RETRO FIRE
- TRANSMITS ELAPSED TIME FROM RETRO FIRE
- 24-HOUR OPERATING LIFE



WEIGHT	20.0 LB
VOLUME	350 CU IN.
POWER	2.5 W
OPERATING FREQUENCY	296.8 MHz

3.0 PRELIMINARY PROGRAM DEFINITION PLAN

The plan provides a fully qualified and flight tested three-man or nine-man escape system for the August 1971 Apollo Applications Program space station launch.

This plan is based upon the system concepts, configuration definition, recommendations and conclusions presented previously. Detailed plans for performing and interpreting for each major program facet are presented in detail in the following sections. The overall program master schedule is presented on the adjoining page. Sufficient preliminary engineering analysis and design will be completed by the third month to permit the release of specifications for procurement of long lead time items.

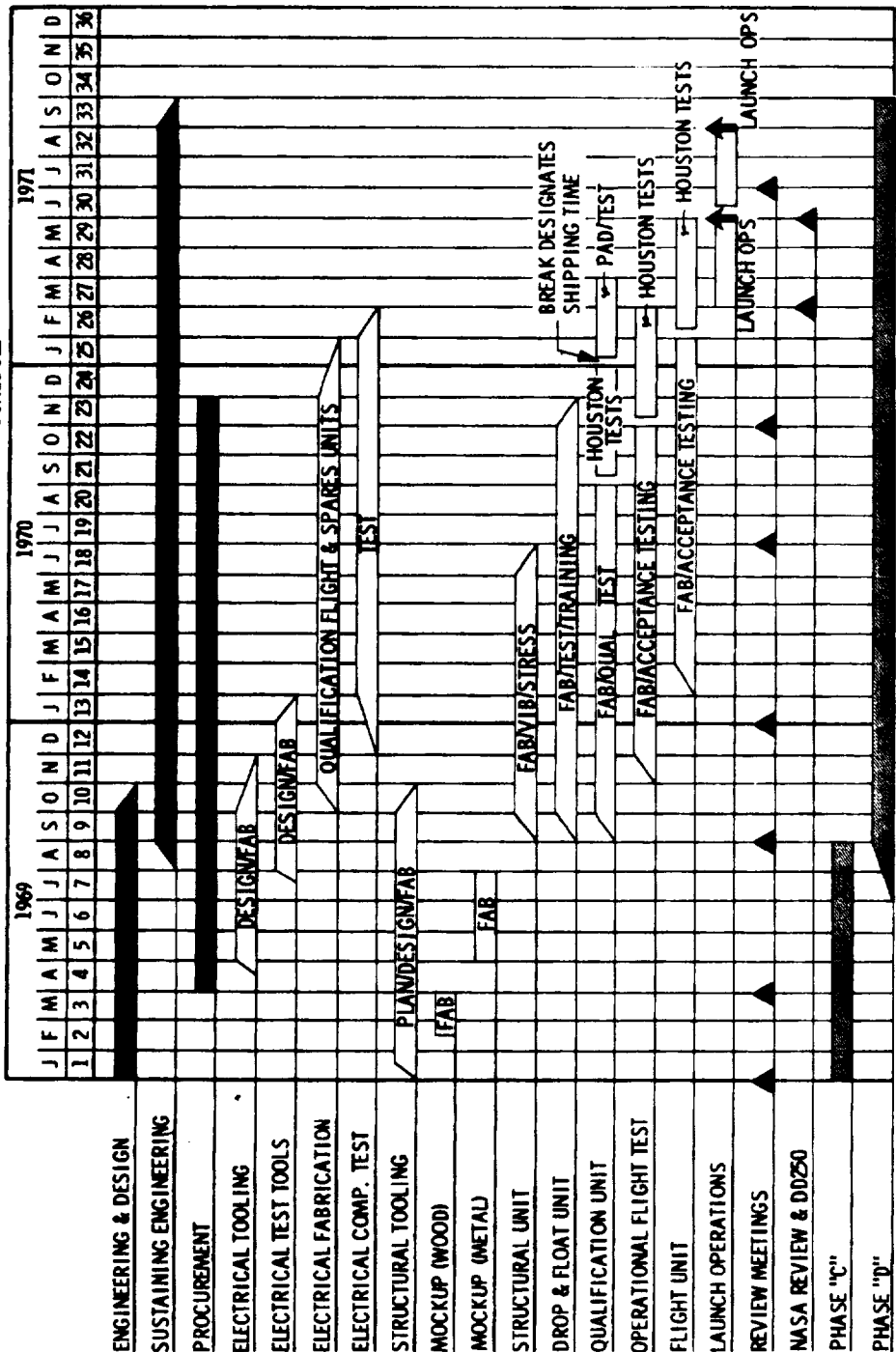
The wooden and metal mock-up aids final design, integration, manufacturing tooling and planning during the initial design phase. The metal and wood mock-up will be used for engineering and manufacturing mock-up of plumbing, electrical cable routing, evaluation of assembly installations and antenna pattern tests. Structural tooling design will be started and planned on the basis of pre-released structural drawings. The structural system presents no special problems. Wind tunnel tests are desired to evaluate the effects of various curvatures of the heatshield and fairing about the cylindrical structural section and should not affect the basic structure assembly or design.

The drop and float test units will contain all required operative equipment required for parachute drop and float test. It is planned that the aircraft drop and float tests be conducted by NASA Manned Space Center, Houston.

The qualification and flight units will undergo manned thermo-vacuum chamber testing at the NASA Manned Space Center, Houston.

The boost vehicle system presents no schedule problems or conflicts and is not presented in order to keep the program plan brief.

ESCAPE SYSTEM PROGRAM PLAN SCHEDULE



3.1 PROGRAM MANAGEMENT PLAN

Lockheed project management is supported in every sector by specialized engineering, scientific, manufacturing and support organizations directed and controlled by the Escape Systems Program Office.

The Phase "C" development of the escape system will receive the specialized engineering talent and scientific specialists required to efficiently analyze, evaluate, design, and integrate the hardware and crew support systems. During the manufacturing test and launch operations phases of the program, the engineering and scientific specialist effort will be only as required to support the program requirements. Lockheed's total pattern of specialized support will adaptively change in accordance with program requirements and status. The ordered processes by which these manifold efforts are directed, coordinated, scheduled, and controlled are central responsibilities of the Program Office.

It is through the Escape System Program Office that NASA Manned Spacecraft Center is ensured direct response to their technical requirements and managerial direction. Continuous and effective liaison between Lockheed and NASA is a prime program office function.

For the last three years Lockheed NASA Programs such as the Lunar Orbiter, EOGO and Nimbus Agena vehicles have all been within or under budget and on schedule.

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3.2 RELIABILITY PROGRAM PLAN

The NASA Programs Product Assurance Organization at Lockheed has an established and effective reliability program plan which meets NASA requirements.

NASA Programs Product Assurance has managed the Quality and Reliability Programs for NASA Agena contracts, such as Ranger, Mariner, Lunar Orbiter, ATS, and the Gemini Agena Target Vehicle.

The following reliability tasks are performed:

- a. Reliability and Safety Analysis of the systems and subsystems, estimates of reliability, and failure mode analysis to identify failure modes and trends.
- b. Engineering document review of drawings, specification, and test procedures.
- c. Review of parts and material applications, human factors and safety provisions.
- d. Component review to determine safety and failure modes due to effect and susceptibility to electrical, thermal, acoustic, pressure and other stresses.
- e. Failure investigation and analysis to determine cause and prevention.
- f. Participation in the Material Review Actions.
- g. Review and approval of test plans, specifications and procedures to ensure completeness and proper definition of NASA requirements.
- h. Monitoring tests and reviewing test results; providing qualification status to program management.
- i. Part selection and application information for designers and for use in the reliability analyses.
- j. Maintenance and review of complete records and history of all parts, components and assemblies used.

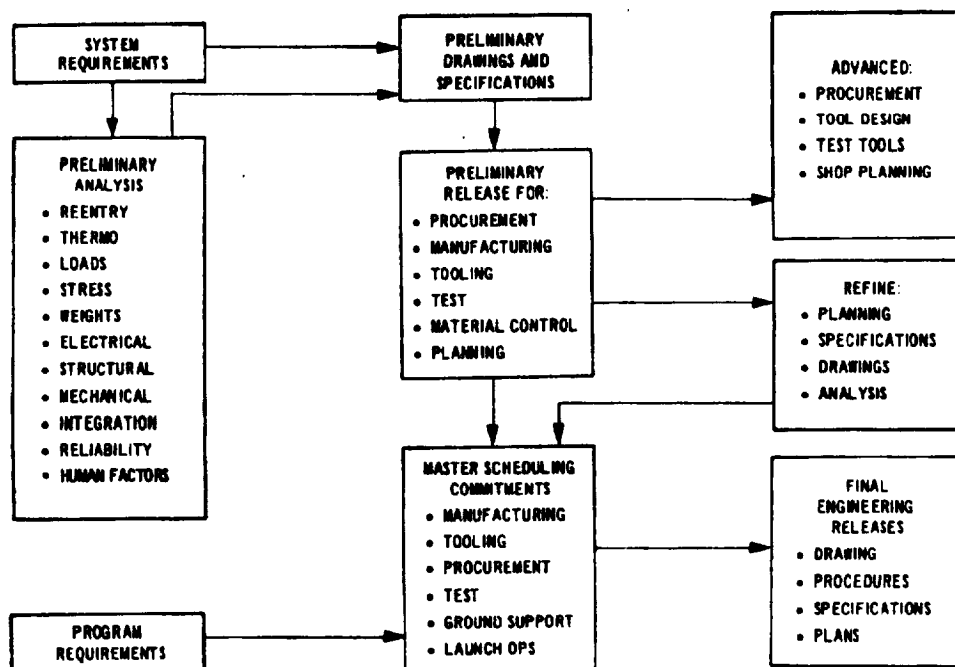
3.3 ENGINEERING AND DESIGN PLAN

The escape system will be designed in ten months.

The program engineering plan is based upon the Phase A and B study results and extensive use of available technology and hardware. Key engineering specialists are readily available to perform technical analysis in aerodynamics, loads and dynamics, stress, re-entry and orbit thermodynamics, preliminary system design and specifications.

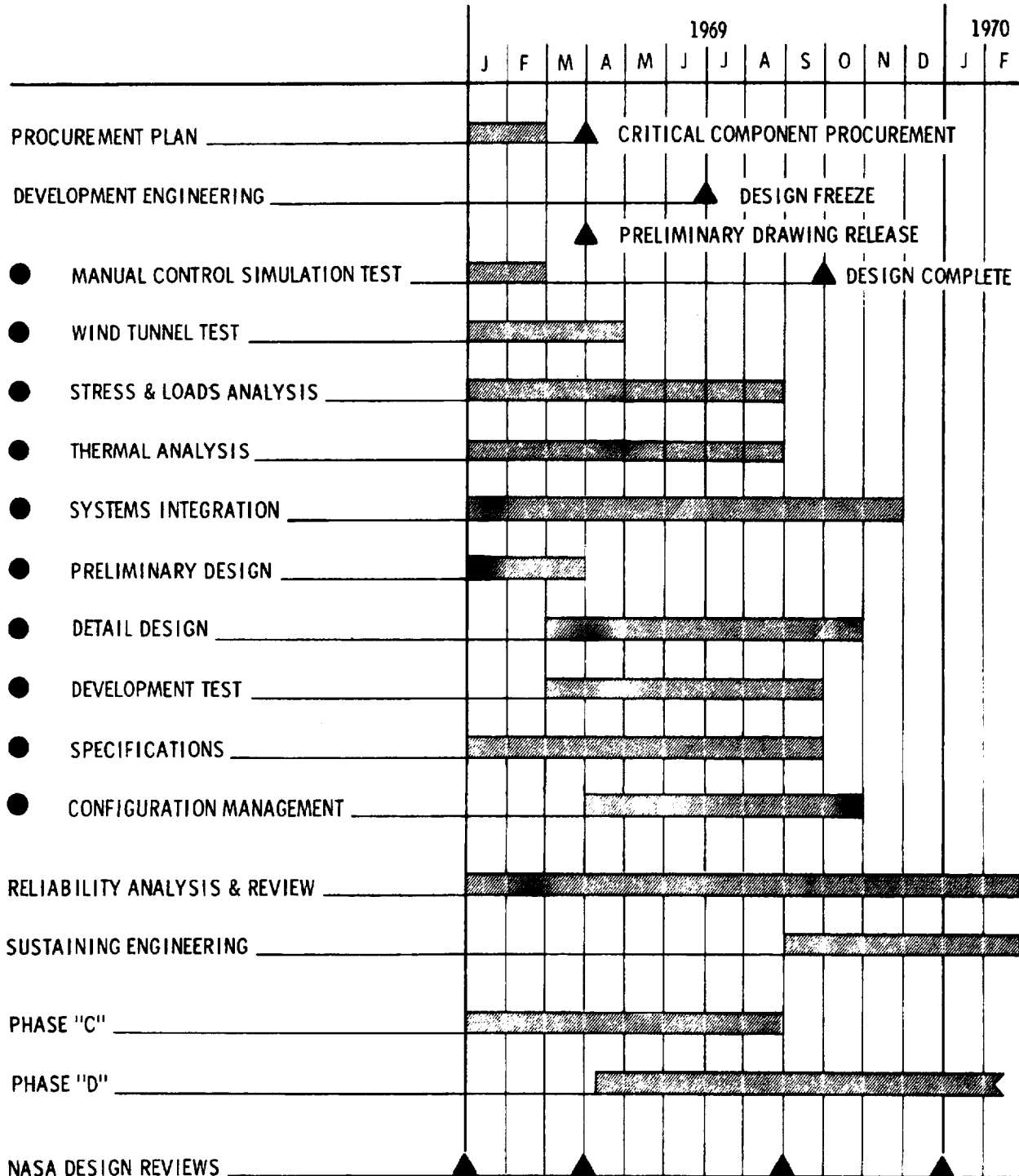
The preliminary engineering and design schedule is shown on the adjoining page. Manufacturing planning and preliminary tool design will be performed from preliminary drawing releases made after the third month from go-ahead. Design freeze and final drawing releases start after the sixth month. Drawing release schedules are coordinate with manufacturing, tooling, and test needs to support the overall program schedule commitments.

Sustaining engineering provides technical direction, analyses and review as required during manufacturing, testing and launch operations.



ENGINEERING SCHEDULE

ENGINEERING AND DESIGN SCHEDULE



3.4 PROCUREMENT PLAN

Components will be purchased from subcontractors producing identical or similar qualified flight components.

The number of components required for assembly of one flight unit are shown. Procurement will include about seven sets of each item as required to provide for component qualification, spares and assembly of the qualification and two flight escape system units. Preliminary procurement analysis indicates that the longest procurement span will be 41 weeks from placement of the order until delivery of qualified components.

<u>RETRO PROPULSION</u>		<u>CONTROL AND DISPLAY</u>	
Motor	4	Sight and window unit	1
Igniter	4	Altimeter	1
Squibs	8	Artificial horizon unit	1
		Clock	1
		Hand controller unit	1
<u>REACTION CONTROL SYSTEM</u>		Switches	as req'd
Hydrazine tank	1	Reentry timer	1
10 lb thrusters	6	Panel lights	1 set
Isolation valves	2	Temp and pressure meters	as req'd
Valve pyros	2	RF power gages	2
Plumbing hardware	as req'd		
<u>COMMUNICATIONS</u>		<u>RECOVERY SYSTEM</u>	
Transceiver	1	Drogue and mortar	1
VHF recovery beacon	1	Main chute	1
Coax switches	2	Flashing light assy.	1
Multiplexer	1	Recovery timer	1
		Barometric switches	2

ENVIRONMENTAL CONTROL

MODULE COMPONENTS

Oxygen tanks	2
30 psi press regulator	2
4.5 psi press regulator	2
Plumbing hardware	as req'd
Compressor	2
Water tanks	2
Valves	4
Fan	1
LiOH/charcoal filter	1
Heat exchanger	1
Isolation valve	1
Pyros	2

ELECTRICAL

Battery	2
Station umbilical	1
Propulsion umbilical	2
Wire	as req'd
Cable	as req'd
Connectors	as req'd

STRUCTURES

Heat shield materials	1
Couch module	3
Pyro squibs	6
Fasteners	as req'd
Structural materials	as req'd
Fittings	as req'd
Docking adapter half	1

3.5 MANUFACTURING PLAN

Escape system qualification and flight units can be subassembled and completed in 28 weeks (140 normal working days).

The following components and assemblies are manufactured by Lockheed:

<u>Mechanical System</u>	<u>Electrical System</u>
Plumbing	Control box
Forward structure dome	Electrical harnesses
Forward cone structure	Antennas
Cylinder structure	Audio unit
Aft cone structure	Orbiting beacon
Hatch and frame assembly	Reaction control electronics
Chute doors	
Docking adapter and cone assembly	
Instrument panel assembly	
Propulsion module and adapter assembly	
Heat shield and fairings	

The manufacturing sequence and schedule are shown on the adjoining page. The structural unit, the first unit manufactured will contain a test heat shield with partial honey-comb. The drop test unit will contain a flight heat-shield. Structural detail parts fabrication will be started for three units at the start of fabrication of the structural test unit.

LMSC-A940555
Volume 4

3.6 TEST PLAN

The test plan provides a full qualified and flight worthy escape system in support of the August 1971 manned AAP flight experiment.

Aerodynamic testing will be performed at NASA wind tunnel facilities to verify aerodynamic stability and sensitivity.

Structural testing is scheduled for initial vibration testing with a search for resonant frequencies before start of the system level qualification testing. Stress testing will verify the integrity and safety margins of the fabricated assemblies. A preliminary drop test into a water tank will be made before the start of qualification testing. Parachute drop tests from an aircraft will be conducted by NASA Manned Spacecraft Center, Houston.

The qualification unit and each flight unit is scheduled for four months at the Manned Spacecraft Center, Houston, for man-rated thermo-vacuum systems test. The qualification unit will be shipped to the Manned Spacecraft Center for final man-rated system qualification testing and then shipped to the Kennedy Space Center for interface tests, a mock countdown and simulated launch.

Each flight unit will be at the launch base for three months prior to launch for interface and fit checks, shroud encapsulation tests and J-FACT (Joint Functional Acceptance Testing).

Each unit is scheduled for a total of twelve months of systems testing in the following sequence:

- Ambient environment system test
- EMI (Electromagnetic Interference) tests
- Vibration test on all three axes
- Acoustic chamber test
- Ambient environmental test

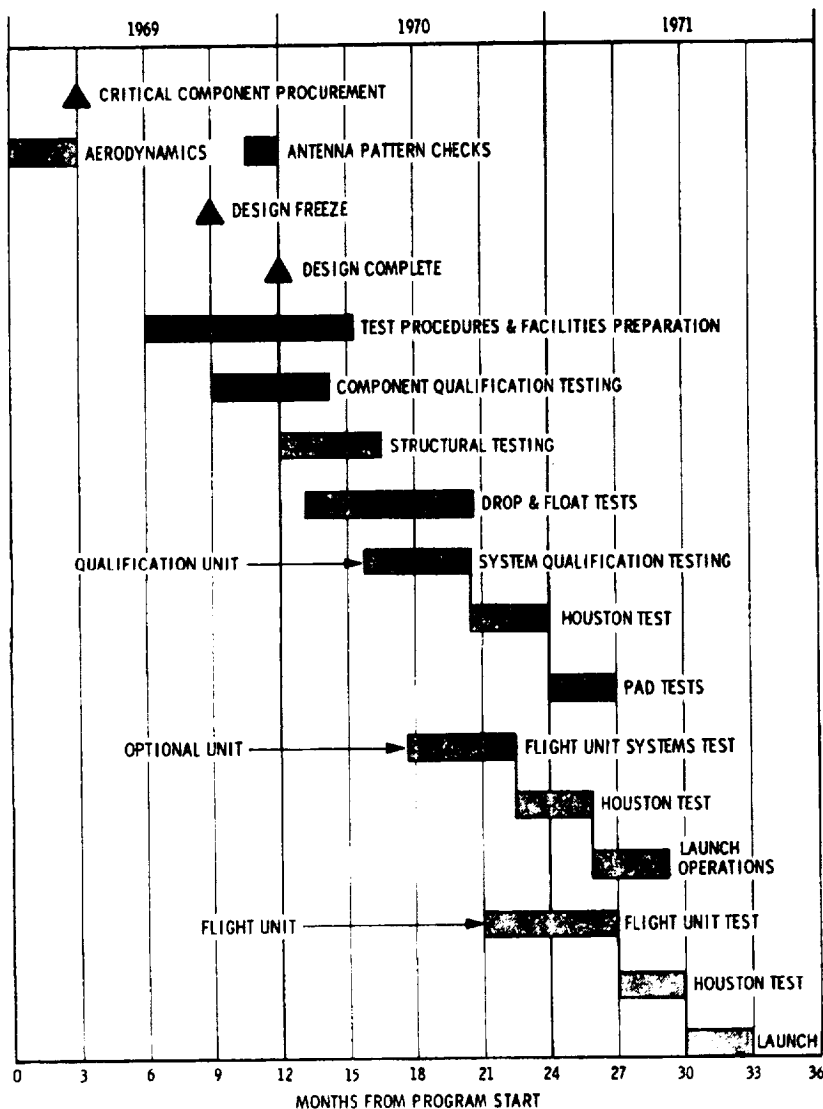
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- Thermo-vacuum system test at Manned Spacecraft Center, Houston
- Launch base tests and Joint Functional Acceptance Tests (J-FACT)

The time span is adequate due to the following factors:

- Extensive experience in operation of the test chambers and testing spacecraft and recovery systems.
- Availability of the facilities and personnel.
- The test sequence and schedule avoid conflict between test units and facilities.

TEST SCHEDULE



3.7 FACILITIES PLAN

Existing facilities are adequate to meet all of the program manufacturing and test requirements.

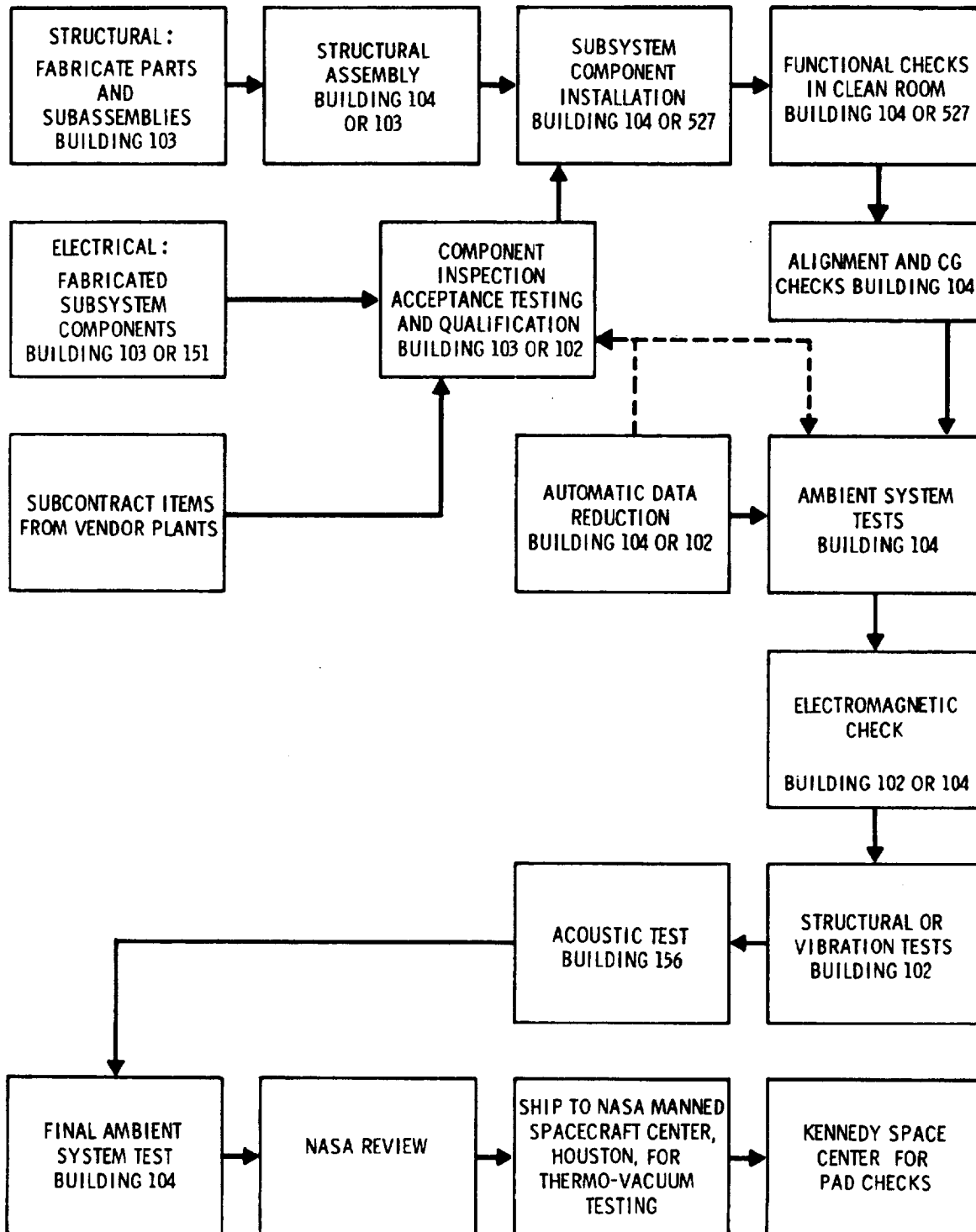
An analysis has been conducted to determine facility requirements for the escape system. The following facility ground rules were assumed for the analysis:

- The three-man system will be over 9 ft diameter by 9 ft long. Basic structure weight – 2500 lb. Gross weight 3500 lb with the three-man crew. The nine-man unit weighs approximately 5200 lb and is over 12 ft by 12 ft in size.
- System will be man-rated with provisions for shirt sleeve environment for a three man crew. A single-gas environmental subsystem using oxygen is employed.
- Escape system will be manufactured and assembled by Lockheed.
- The heat shield will have silicone elastimer injected into each honeycomb cell.
- Final assembly will be performed in a clean-room environment.
- Testing will be performed in accordance with the test plan.

All of the facilities for manufacturing and test of the escape system exist at Lockheed except for the man-rated thermo-vacuum tests which will be conducted at NASA/Houston.

Lockheed Missiles & Space Company's, Sunnyvale California manufacturing facilities include all required manual and numerically controlled machines, manufacturing process and fabrication equipment required for space system work. Some of the larger machines are an 11-foot swing, numerically controlled ring mill and a 12 foot by 30 foot numerically controlled skin mill. Clean rooms for final assembly and test are located in Buildings 104, 159 and 527. Complete component and system test facilities for testing large space systems are available in excess of the requirements outlined by the following facilities flow chart.

MANUFACTURING AND TEST FACILITIES FLOW CHART



3.8 LAUNCH BASE OPERATIONS

The launch base operations plan is based upon the use of available facilities and flight proven methods of launch operations.

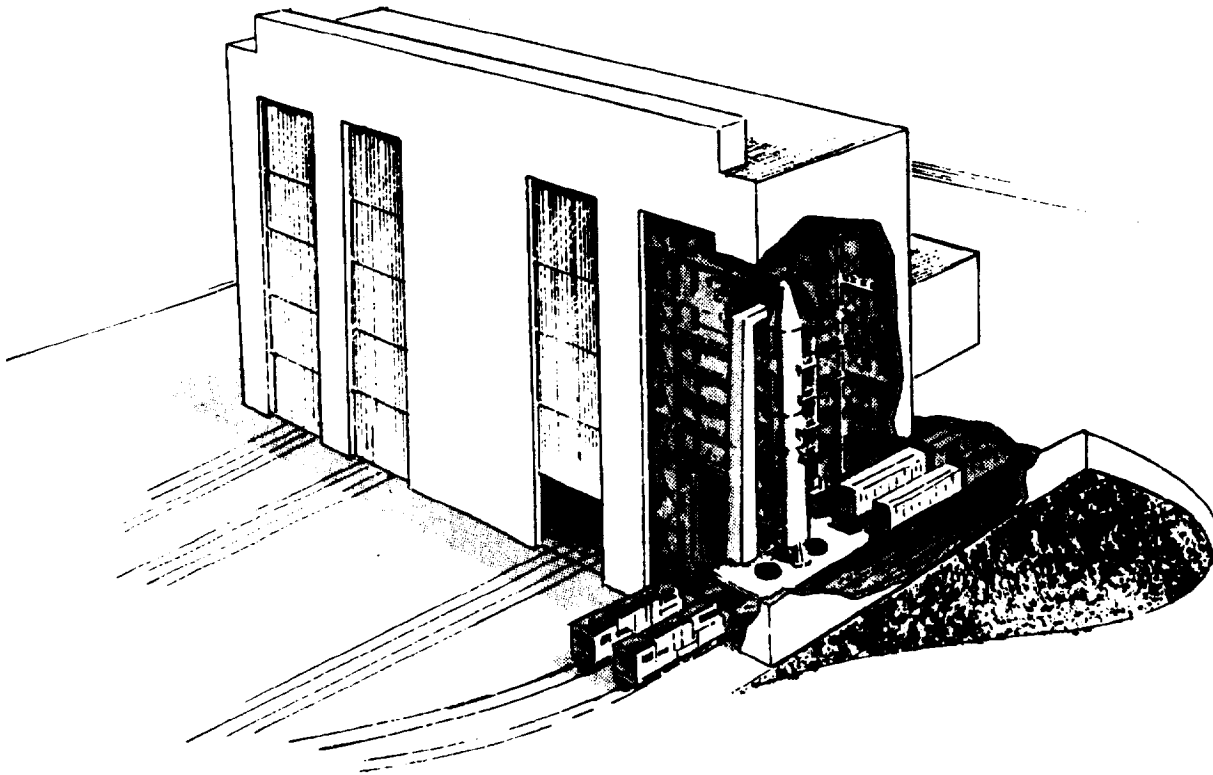
The escape system will be received at the Kennedy Space Center Manned Spaceflight Operations Building in as near flight-ready condition as possible, such that only a minimum of prelaunch system preparation and manual testing of the flight system is required prior to launch.

Prior to testing of the flight escape system at the launch base, a complete compatibility and systems integration test will be performed utilizing the Agena development test vehicle and the qualification escape system. The compatibility tests will check and verify the ground test equipment, the test procedures, and provide test crew training and familiarization. The Integrate, Transfer and Launch facility (ITL Complex 40-41) as planned for the Titan IIIB/Agena Intelsat IV launch in 1970 will provide a proven system test and checkout capability for the complete launch vehicle system. The ITL complex permits most of the following launch operations and test sequence of the complete launch vehicle to be conducted in an enclosed area and with a minimum of mating and handling of the flight vehicles:

1. Receipt of booster stages and escape system.
2. Buildup and alignment of Titan stages I & II and initiation of vehicle and ground system operation checkout.
3. Installation of the Agena vehicle and initiation of operation checkout.
4. Combined Systems Test (CST) of Titan/Agena/qualification escape system with squibs and initiators; simulated countdown; lift-off and flight.
5. Mating of flight escape system and Agena and boost vehicle; escape system integrated testing.
6. Installation of shroud.

7. Final ground-to-vehicle tests.
8. Installation of all pyrotechnics and batteries; removal of gantry tower; loading of propellants, tank pressurization and final preparation for launch.
9. Terminal countdown and launch.

VERTICAL INTEGRATION BUILDING



3.9 PRELIMINARY PRICE ESTIMATES

The AAP escape system is a simple functional unit which provides an economical solution to the escape mission requirements.

An estimate included herein gives the overall cost to the Government for the program outlined in the previous sections of this volume.

The escape system price is based upon estimates of the effort required in the various functional areas to convert the program concept presented to end item flight hardware ready for launch.

The estimates for the various functional areas have been made by responsible Lockheed personnel who specialize in the particular areas involved.

The Titan-Agena and integration and launch estimates are based on LMSC cost/price data for this launch vehicle configuration. In addition, certain additional pricing has been included for program-peculiar items on the Agena needed for the rendezvous and docking maneuver explained in Section 2.0, and for shrouds and adapters.

This pricing has been predicated upon the availability of certain government property and services to the Contractor at no cost. These items are:

- Thermal-vacuum testing
- Airplane drop and float testing
- Wind tunnel testing
- Fuel and supplies at launch base
- Base support at launch base
- Range and tracking support

Launch costs have been estimated based on the assumption that a launch complex configured for Titan-Agena flights will be available from Comsat launches. The fixed cost element of launch operations has been estimated on the basis of 5 to 6 launches per year from this complex.

EMERGENCY EARTH ESCAPE SYSTEM PROGRAM
Price Estimate (Millions)

	<u>3-MAN SYSTEM</u>		<u>9-MAN SYSTEM</u>	
	<u>Non- Recurring</u>	<u>Recurring</u>	<u>Non- Recurring</u>	<u>Recurring</u>
<u>ESCAPE SYSTEM</u>				
Program management and control	\$ 3.23	\$.34	\$ 3.23	\$.34
System engineering and analysis	2.94	.25	2.94	.25
Design engineering	3.01	.48	3.01	.48
Test	3.01	.44	3.05	.44
Hardware fab. & assy.	9.81	1.78	14.23	2.29
Support equipment	<u>3.80</u>	<u>—</u>	<u>4.85</u>	<u>—</u>
TOTAL	<u>\$25.80</u>	<u>\$3.29</u>	<u>\$31.31</u>	<u>\$3.80</u>
<u>TITAN-AGENA</u>				
Agena vehicle	\$ 1.25	\$2.85	\$ 1.25	\$2.85
Agena launch	—	.75	—	.75
Titan IIIB	—	3.45	—	3.45
Titan IIIB launch	—	1.50	—	1.50
Shroud	<u>—</u>	<u>.30</u>	<u>2.50</u>	<u>.40</u>
TOTAL	<u>\$ 1.25</u>	<u>\$8.85</u>	<u>\$ 3.75</u>	<u>\$8.95</u>
PROGRAM TOTALS	<u>\$39.19</u>		<u>\$47.81</u>	